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THE TRANSONIC CHARACTERISTICS OF A LOW-ASPECT-RATIO
TRIANGULAR WING WITH A CONSTANT-CHORD FLAP AS
DETERMINED BY WING-FLOW TESTS, INCLUDING
CORRELATION WITH LARGE-SCALE TESTS

By George A. Rathert, Jr., L. Stewart Rolls,
and Carl M. Hanson

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SUMMARY

Wing-flow-method measurements are presented from 0.47 to 1.15 Mach number of the longitudinal-stability and -control characteristics of a triangular wing of aspect ratio 2 equipped with a constant-chord flap. Comparisons are made with wind-tunnel and free-flight-test results at higher Reynolds numbers to determine to what extent the data are affected by test technique and scale.

Changes in the lift-curve slope and aerodynamic-center position through the transonic speed range were moderate and gradual. The agreement with results from wind-tunnel and free-flight tests was sufficiently good to insure that the aerodynamic characteristics presented for this plan form are qualitatively valid.

The flap retained positive effectiveness and nearly linear lift and pitching-moment characteristics at all speeds and Mach numbers. The over-all loss in lift effectiveness, about 30 percent, at transonic speeds and the lift effectiveness at supersonic speeds were in relatively good agreement with the larger-scale wind-tunnel and free-flight tests. The variation of lift effectiveness with Mach number at high subsonic speeds, however, was not in agreement, possibly due to operation of the flap in an increasingly thick separated flow region. The variation of hinge moment with flap angle increased considerably at transonic speeds. The abruptness of the increase was a function of test technique and scale but the change was large enough to constitute a design problem.

INTRODUCTION

Low aspect ratio, large amounts of leading-edge sweep, and extreme taper are plan-form characteristics increasingly associated with lifting or control surfaces for missiles and highly maneuverable aircraft where design speeds are moderately supersonic and operation in the transonic speed range is required. Information on this type of plan form is being furnished through a coordinated study of a particular triangular wing of aspect ratio 2 by several facilities of the Ames Aeronautical Laboratory. Results at low speeds have been reported in references 1, 2, and 3, at moderate to high subsonic speeds in references 4 and 5, and at supersonic speeds, 1.09 to 1.53 Mach number, in references 5 and 6. The characteristics of a constant-chord flap fitted to the basic wing are presented at low speed and full scale in reference 1 and up to high subsonic speeds in reference 7.

This report adds the transonic longitudinal-stability and -control characteristics indicated by wing-flow-method measurements extending continuously through the Mach number range 0.47 to 1.15. Some hinge-moment data and the effectiveness of the constant-chord flap are included.

The validity of the wing-flow-method measurements has been investigated by comparing them with results from the above-mentioned references. Comparisons also have been made with results from investigations of models of a complete fighter airplane which differed in detail from the subject configuration but had a comparable triangular wing and constant-chord flap. Large- and small-scale tests of this configuration at transonic speeds are presented in references 8 and 9, respectively.

COEFFICIENTS AND SYMBOLS

The following coefficients and symbols are used in this report:

C_h hinge-moment coefficient $\left(\frac{\text{hinge moment}}{q b_f \bar{c}_f^2} \right)$

C_L lift coefficient $\left(\frac{\text{lift}}{qS'} \right)$

C_m pitching-moment coefficient about the quarter-chord point of the wing mean aerodynamic chord $\left(\frac{\text{pitching moment}}{qS' \bar{c}} \right)$

C_{h_α} rate of change of hinge-moment coefficient with angle of attack, per degree

- C_{h_δ} rate of change of hinge-moment coefficient with flap angle, per degree
- C_{L_α} rate of change of lift coefficient with angle of attack, per degree
- C_{L_δ} rate of change of lift coefficient with flap angle, per degree
- C_{m_α} rate of change of pitching-moment coefficient with angle of attack, per degree
- C_{m_δ} rate of change of pitching-moment coefficient with flap angle, per degree
- M Mach number $\left(\frac{V}{a}\right)$
- R Reynolds number $\left(\frac{\rho V \bar{c}}{\mu}\right)$
- S' wing area of the semispan model, square feet
- V airspeed, feet per second
- a speed of sound, feet per second
- b' wing semispan, feet
- b_f span of the flap, feet
- c local chord, feet
- $\bar{c}$ wing mean aerodynamic chord $\left(\frac{\int_0^{b'} c^2 dy}{S'}\right)$, feet
- c_f chord of the flap, feet
- $\bar{c}_f$ root-mean-square chord of the flap aft of the hinge line, feet

- q dynamic pressure $\left(\frac{1}{2}\rho v^2\right)$, pounds per square foot
- y spanwise location, feet
- α angle of attack, degrees
- δ angle of the flap relative to the wing chord line (corrected for distortion under aerodynamic load), degrees
- δ_u flap angle (uncorrected for distortion effects), degrees
- μ air viscosity, slugs per foot-second
- ρ mass density of air, slugs per cubic foot

TEST EQUIPMENT

Flow Field

The data were obtained by mounting semispan models in a region of accelerated air flow over a built-up test station on an airplane wing. The test Mach numbers presented are those determined without the model in place but at a location corresponding to the model centroid of area. Due to pressure gradients in the flow field, the Mach number decreased approximately 0.01 per inch spanwise from the root to the tip of the model. Chordwise, the Mach number gradient was zero at low speeds and increased uniformly to a value of 0.01 per inch, increasing from apex to trailing edge, at a Mach number of 1.15. The ratio of the displacement thickness of the test-station boundary layer to the semispan of the model was 0.011. The flow field measurements are discussed more completely in reference 10.

Models

The dimensions of the steel semispan models are presented in figure 1. The basic plan form was a triangle of aspect ratio 2, leading-edge sweep, 63.4° . The airfoil section was a symmetrical double wedge with a maximum thickness ratio of 5 percent. Three models were used: one with maximum thickness at 20-percent chord, a duplicate with a constant-chord plain flap $\left(\frac{\text{flap area}}{\text{wing area}} = 0.20\right)$, and one with maximum thickness at 50-percent chord.

The inboard end of the flap was clamped rigidly in a cylinder which could be rotated to vary the flap angle. Spanwise support was provided by the two hinges shown in figure 1. The average gap between the flap and the wing was 0.005 inch or 0.006 flap-chord length. Figure 2 is a photograph showing the flap deflected.

The model with maximum thickness at 20-percent chord and no flap was also tested with two modifications. As a change in the surface condition, the first 20 percent of the chord was coated with a mixture of lampblack and lacquer containing particles 0.002 inch high; also the original ridge line was rounded sufficiently to reduce the section thickness ratio from 5.0 to 4.8 percent.

Balances

Two types of recording balance were used. The lift and pitching-moment characteristics were measured on a three-component electric-strain-gage force balance which was rotated at a rate of 1° per second to vary the model angle of attack. This equipment has been described in reference 10.

The flap hinge-moment data were determined separately with a second balance upon which the flap was oscillated at a rate of 4° per second while the flap angle and hinge moment were recorded. The forward portion of the model wing was fixed to the test-station surface at approximately 0° angle of attack with respect to the local air flow. The hinge moment was recorded by photographing a dial gage attached to a beam restraining the flap in torsion.

DATA CORRECTIONS AND ACCURACY

Corrections derived from static calibrations were applied to compensate for the effects on the angle of attack and flap angle of strain-gage deflection and flap distortion under load. The deflections of the wing itself under maximum load were small and effects on the test data negligible.

Since the forward portion of the model was fixed during the hinge-moment tests, variations recorded in the local flow angle (due to airplane yaw or similar effects) produced a hinge-moment increment proportional to C_{h_α} . These flow angle changes were held over all to $\pm 1.0^\circ$ and were relatively constant within one oscillation of the flap. Thus, while there was some scatter in the hinge-moment coefficients at a flap angle of 0° , C_{h_δ} was not affected and no corrections were applied.

The accuracy of the physical measurements, including the effects discussed, is summarized by the following table presenting the uncertainty in representative values of the test data at a lift coefficient of 0.50 for the highest and lowest dynamic pressures. The listed uncertainties are due principally to factors such as strain-gage zero drifts which remain constant within one oscillation of the model; therefore, slope parameters such as C_{L_α} and C_{m_α} are considerably more accurate than the tabulated absolute values or intercepts.

Quantity	Absolute value and uncertainty	
	M = 0.47 q = 172 lb/sq ft	M = 1.15 q = 697 lb/sq ft
Mach number, M	0.47 ± 0.01	1.15 ± 0.02
Angle of attack α , degrees	11.5 ± 0.4	10.1 ± 0.4
Lift coefficient C_L	0.500 ± 0.035	0.500 ± 0.014
Pitching-moment coefficient C_m	-0.062 ± 0.006	-0.112 ± 0.001
¹ Flap angle δ , degrees	7.8 ± 0.3	7.1 ± 0.3
¹ Hinge-moment coefficient C_h	-0.050 ± 0.018	-0.161 ± 0.069

¹Computed for $\alpha = 0^\circ$ and $\delta_u = 8^\circ$.

TESTS AND RESULTS

The lift and pitching-moment forces on the complete models were recorded from -4° to $+16^\circ$ angle of attack at various constant Mach numbers from 0.47 to 1.15. The corresponding test Reynolds numbers, based on model mean aerodynamic chord, varied from 920,000 to 1,690,000 as shown in figure 3. The following configurations were tested:

- Maximum thickness at 20-percent chord, sharp ridge line, polished surface, no flap
- Maximum thickness at 20-percent chord, sharp ridge line, forward 20 percent of surface coated with lampblack, no flap
- Maximum thickness at 20-percent chord, rounded ridge line, polished surface, no flap
- Maximum thickness at 50-percent chord, sharp ridge line, polished surface, no flap
- Maximum thickness at 20-percent chord, sharp ridge line, polished surface, constant-chord flap; δ_u (gap unsealed) = -4° , 0° , 1° , 2° , 4° , 6° , 8° , 10° , 20° ; δ_u (gap sealed) = 8°

The lift and pitching-moment characteristics of the basic wing (configuration (a)) at various Mach numbers are presented in figure 4. For the other configurations the data are presented only in the form of cross plots which illustrate the significant effects of the various modifications. The effects of the airfoil-section and surface-condition modifications, configurations (b) through (d), are shown at three representative Mach numbers in figure 5.

The flap-effectiveness measurements (fig. 6) are presented in the form of cross plots showing the variations of lift and pitching-moment coefficients with flap angle at an angle of attack at 0° . Symbols are presented in the figure in order to show the number of points used to determine the fairing and the amount of scatter present. Figure 7 shows at three representative Mach numbers the small effect of angle of attack up to 8° and the effect of sealing the gap between the flap and the wing.

The flap hinge moments were recorded from -15° to $+15^\circ$ flap angle at an angle of attack of 0° . These data are presented in figure 8. The amount of displacement of the curves corresponding to increasing and decreasing flap angle is an indication of the amount of friction and hysteresis.

DISCUSSION

Data obtained by the wing-flow method can be used with added confidence if their validity is substantiated by comparison with results from more conventional test techniques. In the subsequent discussion, agreement between the wing-flow data and data obtained at relatively large scale from wind-tunnel and free-flight tests of comparable models will be used to establish which of the wing-flow results can be applied usefully to full-scale configurations and which would not be valid. In the latter case, the differences which will be noted may be due either to deficiencies of the test method or to aerodynamic scale.

The comparisons are presented in two stages, differing in the closeness with which the models and test configurations match those of the wing-flow tests. The primary correlation is with the Ames 12-foot pressure wind-tunnel results reported in references 4 and 7, which are for a model of the identical plan form, airfoil section, and flap configuration used in the wing-flow tests. Semispan test configurations were used in both cases and the ratios of the test-station boundary-layer-displacement thickness to model height were similar, 0.014 for the wind-tunnel tests and 0.011 for the wing-flow tests. For the Mach numbers for which the wind-tunnel data are available then, the comparison would show primarily the combined effects in the wing-flow tests of the nonuniform flow field and of the lower scale.

The secondary comparisons are much more general and include data both for full-span, sting-mounted, wind-tunnel models of the same wing and for a complete triangular-wing fighter-airplane configuration. The latter had a slightly greater aspect ratio, 2.31 compared to 2.00; a slightly greater thickness ratio, 0.06 compared to 0.05; the same percent-area flap; but had a rounded leading-edge airfoil section and a large fuselage and vertical tail. Data from free-flight rocket-powered models and a sting-mounted wind-tunnel model of this configuration have been used primarily to furnish comparisons of control characteristics to higher Mach numbers than are available from the tests of reference 7.

Triangular-Wing Characteristics

Basic wing.— Results obtained directly from the wing-flow investigation of the basic wing, figure 4, show no lift-coefficient or moment-coefficient irregularities or force breaks within the ranges of Mach number and angle of attack investigated. The lift-curve slope and aerodynamic-center location are presented in figure 9 along with the corresponding Ames 12-foot pressure wind-tunnel data for Mach numbers up to 0.95 from reference 4. In the wing-flow tests, the lift-curve slope increases smoothly to a maximum of 123 percent of the low-speed value, then decreases at supersonic speeds. The pitching-moment-curve slopes indicate gradual rearward movement of the aerodynamic center from 0.38 $\bar{c}$ at 0.50 Mach number to 0.47 $\bar{c}$ at 1.15 Mach number.

The effects of Mach number on the lift-curve slope and aerodynamic-center position and the increase in lift-curve slope at high lift coefficients indicated by the wing-flow and the wind-tunnel tests are similar, although at a given angle of attack the wing-flow model develops less lift and has a more forward center of pressure. The differences, 6 percent of C_{L_α} and 0.02 $\bar{c}$, are relatively small and unaffected by Mach number. This comparison is supported by the more extensive but less exact secondary correlation with experimental results and theory shown in figure 10. General agreement exists between the various test techniques. A point of particular interest is the utility of the theoretical methods of references 11, 12, and 13 at subsonic, sonic, and supersonic speeds, respectively. The data in figure 10 emphasize the absence of extreme changes in longitudinal stability at transonic speeds, one of the most desirable characteristics of this plan form.

All-movable control surface.— The data of figures 9 and 10 also indicate advantages for the low-aspect-ratio triangular plan form as an all-movable control surface where it is necessary to operate in or traverse the transonic speed range. The effectiveness is shown by the lift-curve slope (fig. 9), while the hinge-moment characteristics are determined by the variation of pitching-moment coefficient with angle of attack C_{m_α} shown in figure 11 as a function of Mach number. This type of control

surface would provide a considerable amount of effectiveness at all speeds with relatively moderate hinge-moment changes. These characteristics are particularly important where the size and weight of the control actuators or the total amount of energy available are paramount considerations.

Airfoil-section modifications.— Two different locations of maximum thickness, 20-percent and 50-percent chord, were tested to determine if varying the location to allow for a thicker nose section and a larger leading-edge radius or to improve the supersonic drag characteristics (references 14 and 5) would affect the transonic stability characteristics. Figures 5 and 12 indicate little effect except a slightly greater movement rearward of the aerodynamic center with increasing Mach number in the case of the model with the 50-percent-chord location of maximum thickness.

The ridge-line and surface-condition modifications were intended to give some additional insight into the effects of low Reynolds numbers on this plan-form and airfoil-section combination. Viscous effects could change the surface-pressure distributions at this scale by affecting the pressure peak over the physically sharp ridge-line, transition, and flow-separation characteristics. It was hoped that these effects might be simulated by the modifications. The data in figure 12 show, however, that neither rounding the ridge line nor adding surface roughness had any significant qualitative effect. Rounding the ridge line did decrease the lift-curve slope about 7 percent, contrary to the results of reference 4 where a similar amount of rounding had no effect at the larger scale.

Flap Characteristics

Effectiveness.— The flap characteristics in figures 6 and 7 are nearly linear functions of flap angle and angle of attack at all Mach numbers, even at flap angles of 1° and 2° . The comparison in figure 7 between the wing-flow results and the Ames 12-foot pressure wind-tunnel data from reference 7 indicates that at the larger scale the flap produced proportionally greater lift at the lower flap angles, resulting in nonlinear characteristics, particularly at high subsonic speeds.

The effect of sealing the $0.006c_f$ gap is also shown in figure 7 at uncorrected flap angles of 0° and 8° . Sealing the gap increased the lift effectiveness about 9 percent, but did not change the variation with Mach number and therefore was not investigated further.

In figure 13, the flap-effectiveness parameters $C_{L\delta}$ and $C_{m\delta}$, derived from figure 6 over a flap-angle range of -4° to $+4^\circ$, are compared with the corresponding large-scale values up to a Mach number of 0.95 from reference 7. In the wind-tunnel tests, the gap between the flap and the wing was $0.004c_f$ compared to $0.006c_f$ for the wing-flow tests.

Although the lift-effectiveness values agree fairly well at low speeds, the trends with increasing subsonic Mach number are contradictory. The Ames 12-foot pressure wind-tunnel tests indicate a continuous increase in effectiveness to 0.95 Mach number, the limit of the tests. The wing-flow data show no increase; instead the flap loses effectiveness gradually from 0.70 to 1.05 Mach number, the loss being about 30 percent. The difference is greater than could reasonably be attributed simply to the "softening" effect of the Mach number gradients in the flow field.

When the secondary comparisons with control-effectiveness data from references 8 and 9 are considered (fig. 14), the same differences between small- and large-scale results are noted. The two sets of small-scale lift-effectiveness data agree qualitatively as do the two sets of large-scale data, despite differences in test technique in each case, indicating that the lift effectiveness is definitely subject to a considerable scale effect. From the previous consideration of figure 7, it appears that the effect at low Reynolds number shows up primarily as a reduction in the effectiveness at small flap angles, suggesting that at small scale the flap may be operating in a separated flow region increasing in thickness with increasing Mach number. Both the magnitude of the over-all loss in effectiveness at transonic speeds and the values at supersonic speeds are relatively unaffected.

The effects of Mach number on the center of pressure of the additional loading due to the deflection of the flap are summarized in figure 15. Here the difference between the large-scale wind-tunnel tests and the wing-flow data is comparatively small (0.04c), indicating that the distribution of the additional load due to the flap on the wing-flow model is relatively unaffected by the small scale even though the total amount is reduced. The shift of the center of pressure of the additional loading back toward the center of area of the flap at supersonic speeds is shown.

The significance of the effectiveness measurements in terms of the ability of the constant-chord flap to balance a hypothetical all-wing airplane of this plan form through the transonic speed range is demonstrated in figure 16. Level flight at a pressure altitude of 30,000 feet, a wing loading of 60 pounds per square foot, and a center-of-gravity position of 0.30c to provide the same static margins as the similar example from reference 7 have been assumed. A total control-surface movement of 2.2° is required as the Mach number is increased from 0.50 to 1.15 and no abrupt deflections are required.

Hinge-moment characteristics.— Figure 17 shows the effect of Mach number on the hinge-moment parameter C_{h8} derived from the wing-flow tests (fig. 8). Both sets of comparable wind-tunnel and free-flight data (references 7 and 8) are included. A considerable increase in C_{h8} as supersonic speeds are approached is apparent. At a Mach number of 1.05 C_{h8} is 300 percent of the low-speed value, but the absolute value and the rate of the increase with Mach number are functions of test technique and

scale. The general qualitative trend of the change shown at transonic speeds by the wing-flow tests is in agreement with the results from the rocket-powered models. As pointed out in reference 7, this increase in C_{hs} is a serious control-design problem and figure 17 indicates large-scale tests would be essential for quantitative information on a particular configuration.

CONCLUSIONS

Wing-flow-method measurements from 0.47 to 1.15 Mach number at Reynolds numbers from 920,000 to 1,690,000 of the characteristics of an aspect ratio 2 triangular wing with a constant-chord flap and comparisons with larger-scale wind-tunnel and free-flight data have resulted in the following conclusions:

1. Changes in lift-curve slope and aerodynamic-center position at transonic speeds were moderate and gradual. Agreement with the larger-scale tests was sufficiently good to indicate that these characteristics are qualitatively valid.
2. The constant-chord flap retained positive effectiveness and nearly linear lift and pitching-moment characteristics throughout the Mach number and flap-angle ranges. In the wind-tunnel and free-flight tests the flap was proportionally more effective at small flap angles, resulting in non-linear characteristics.
3. Approximately 30 percent of the flap lift effectiveness was lost at transonic speeds; however, the change was gradual and computations showed that a hypothetical all-wing airplane could be balanced from 0.50 to 1.15 Mach number without excessive or abrupt control movements. The variation of lift effectiveness with Mach number at high subsonic speeds was opposite to that noted in the wind-tunnel tests, possibly due to operation of the flap in an increasingly thick separated flow region. The magnitude of the over-all loss at transonic speeds and the values at supersonic speeds showed relatively satisfactory agreement.
4. At a Mach number of 1.05, the rate of change of hinge-moment coefficient with flap angle at $\alpha=0^\circ$ was almost 300 percent of the subsonic value, although the abruptness of the increase was again a function of test technique and scale.
5. The low Reynolds number semispan wing-flow test technique provided a satisfactory basis for qualitative analysis at transonic speeds, but the control-effectiveness and hinge-moment data are not applicable to full-scale conditions, particularly at high subsonic speeds.

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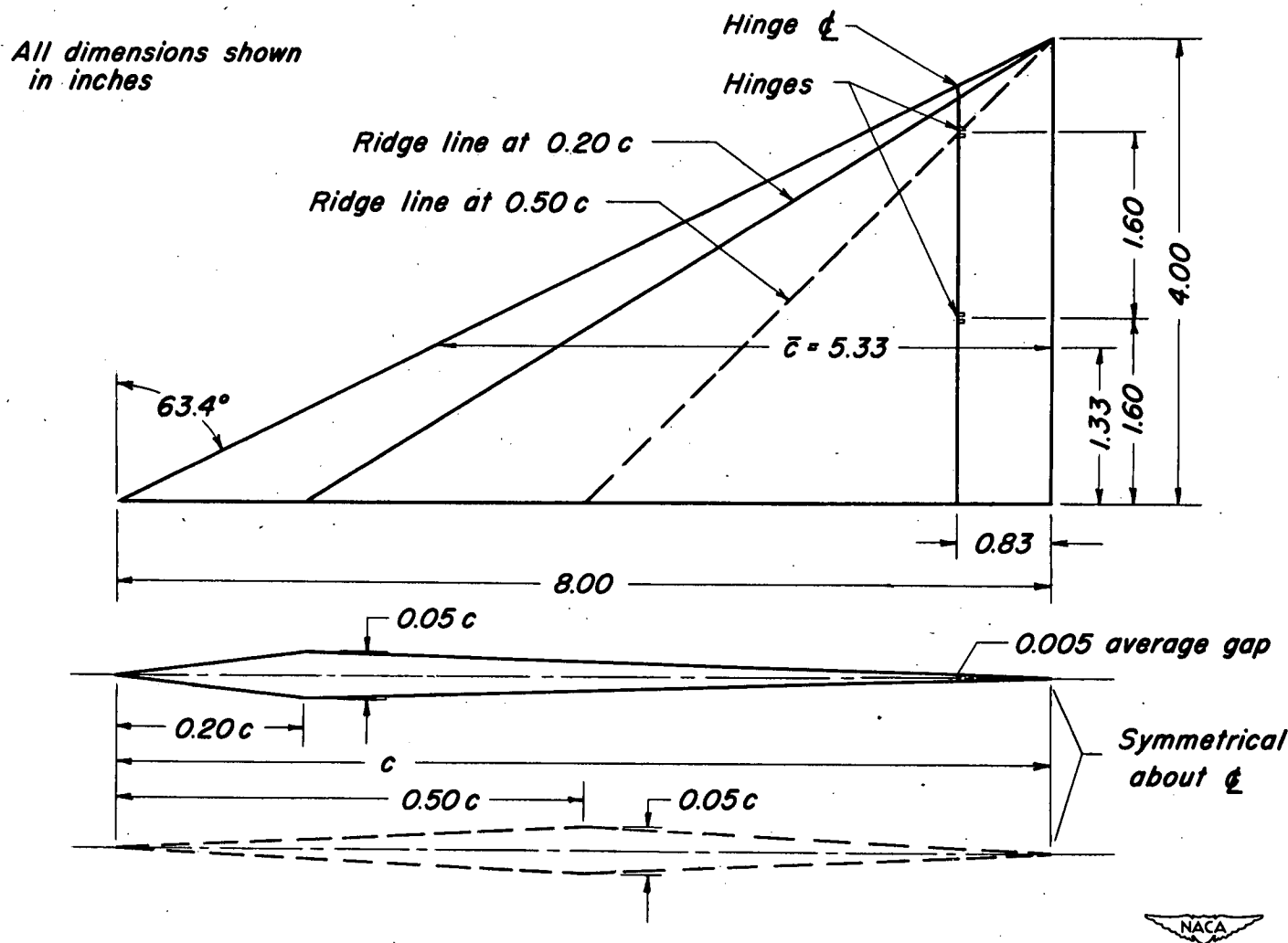


Figure 1.- Dimensions of the models showing the constant-chord flap and two locations of maximum thickness.

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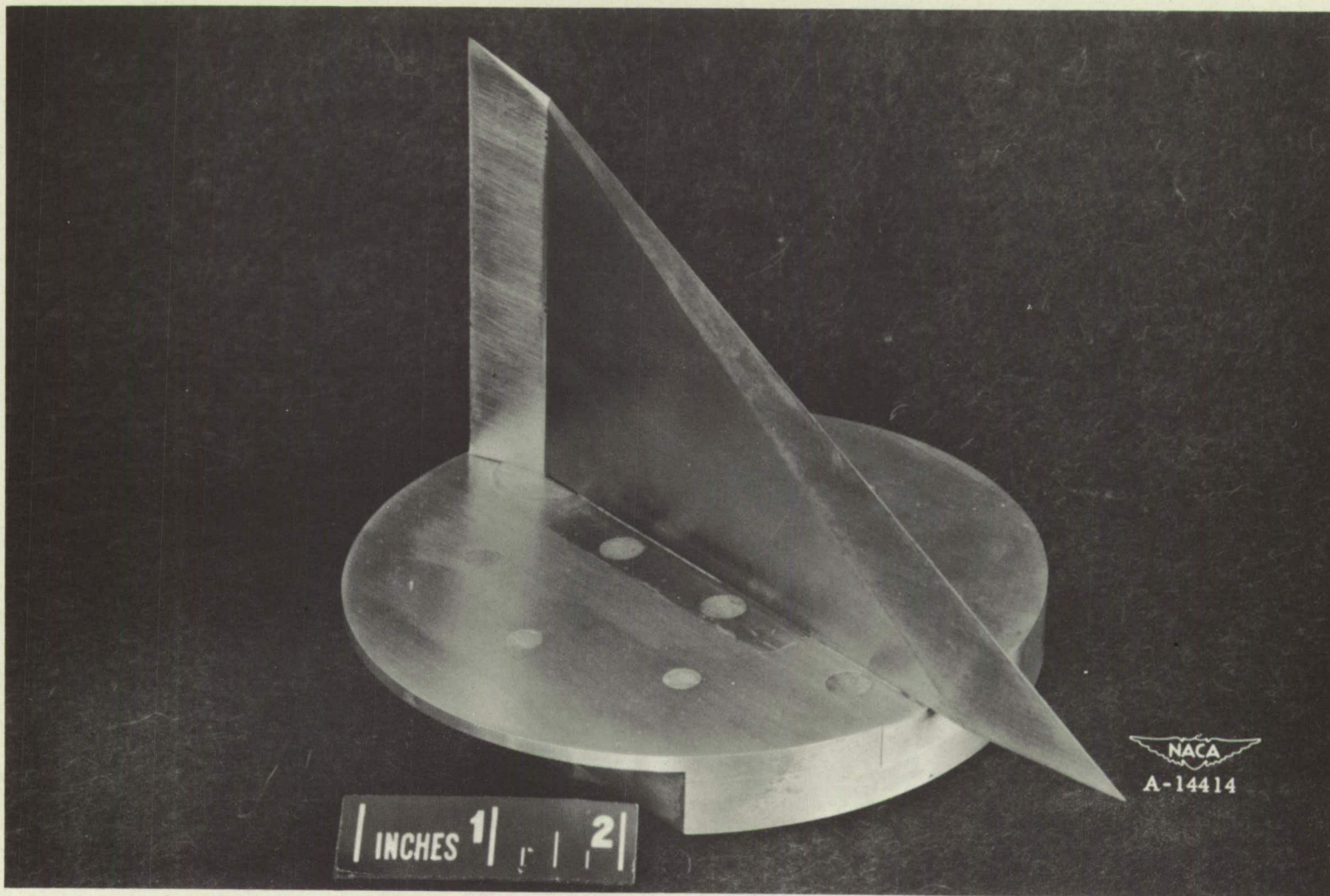


Figure 2.— The triangular wing model with the constant-chord flap deflected, maximum thickness at 0.20 chord, sharp ridge line, polished surface.

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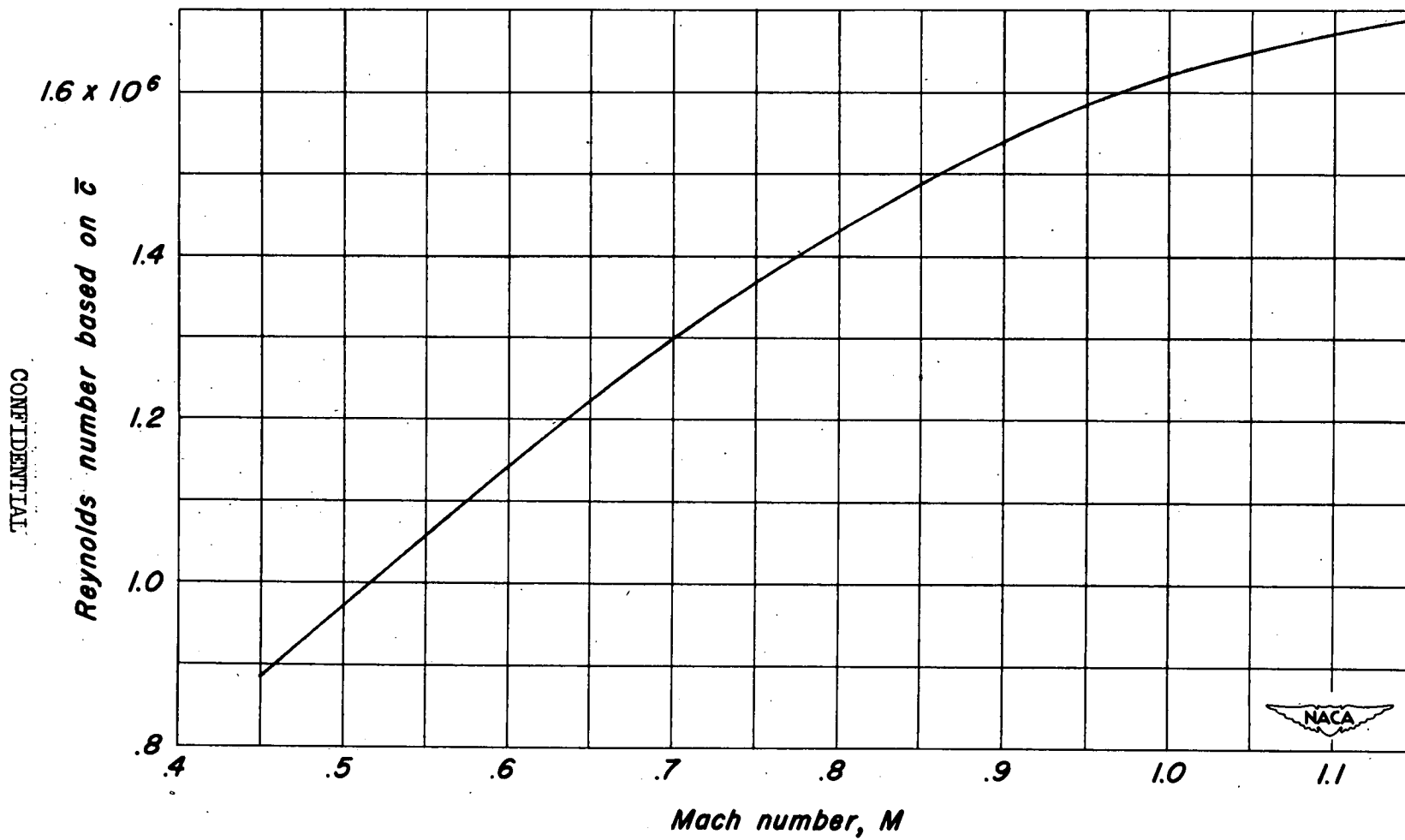
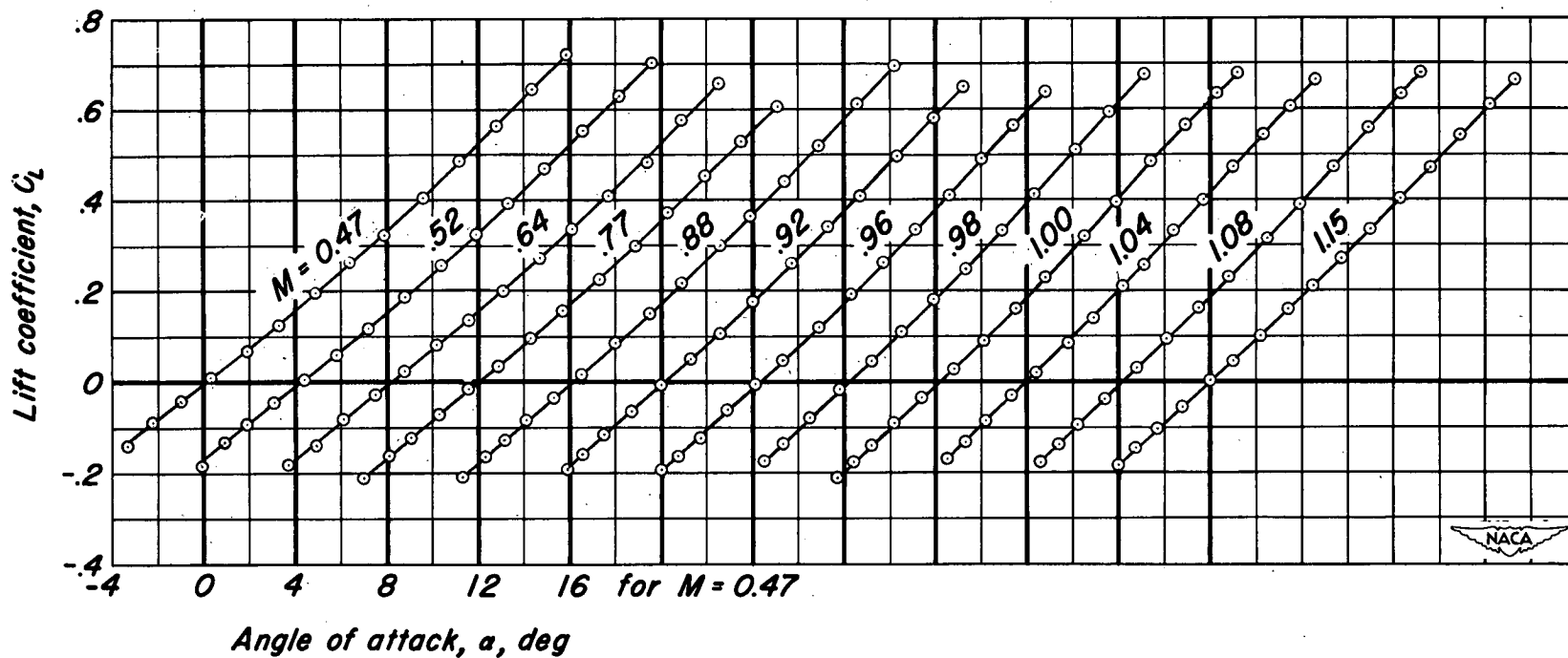
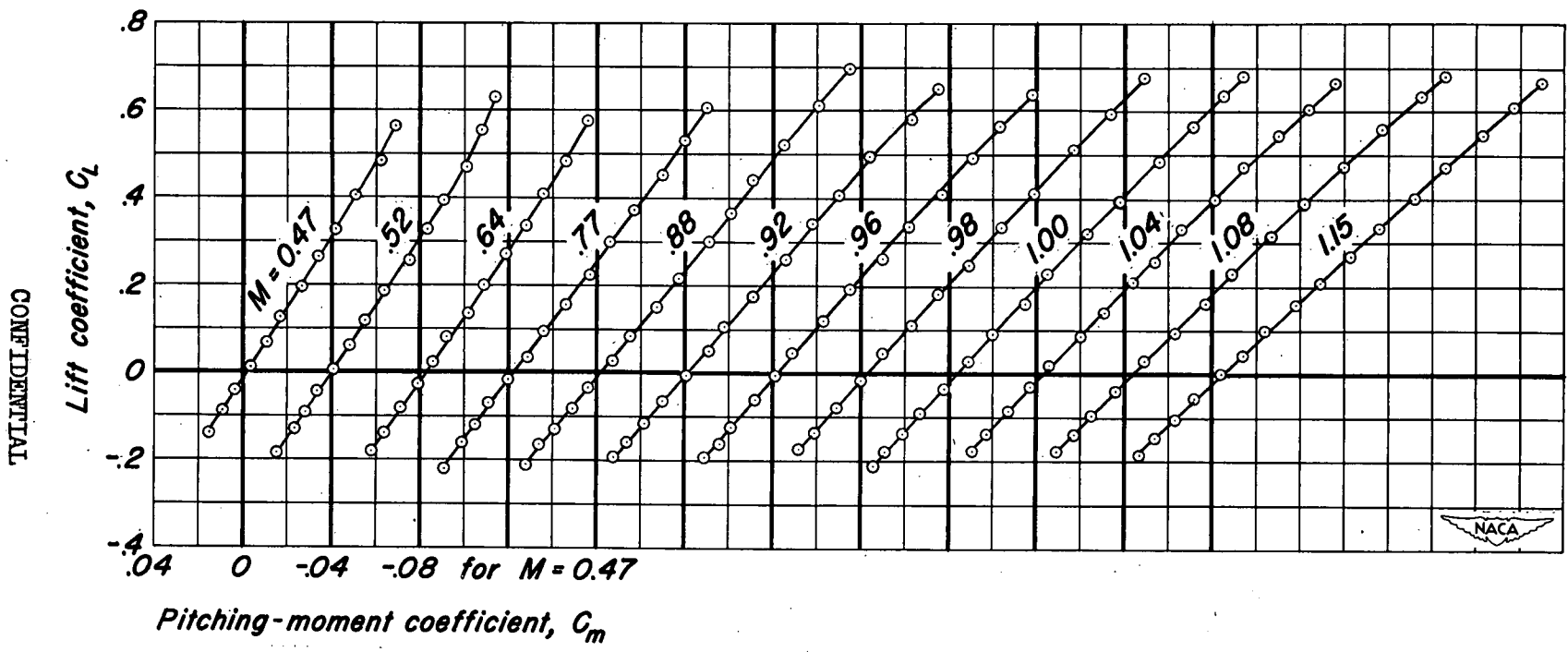


Figure 3.- Variation of test Reynolds number with Mach number.



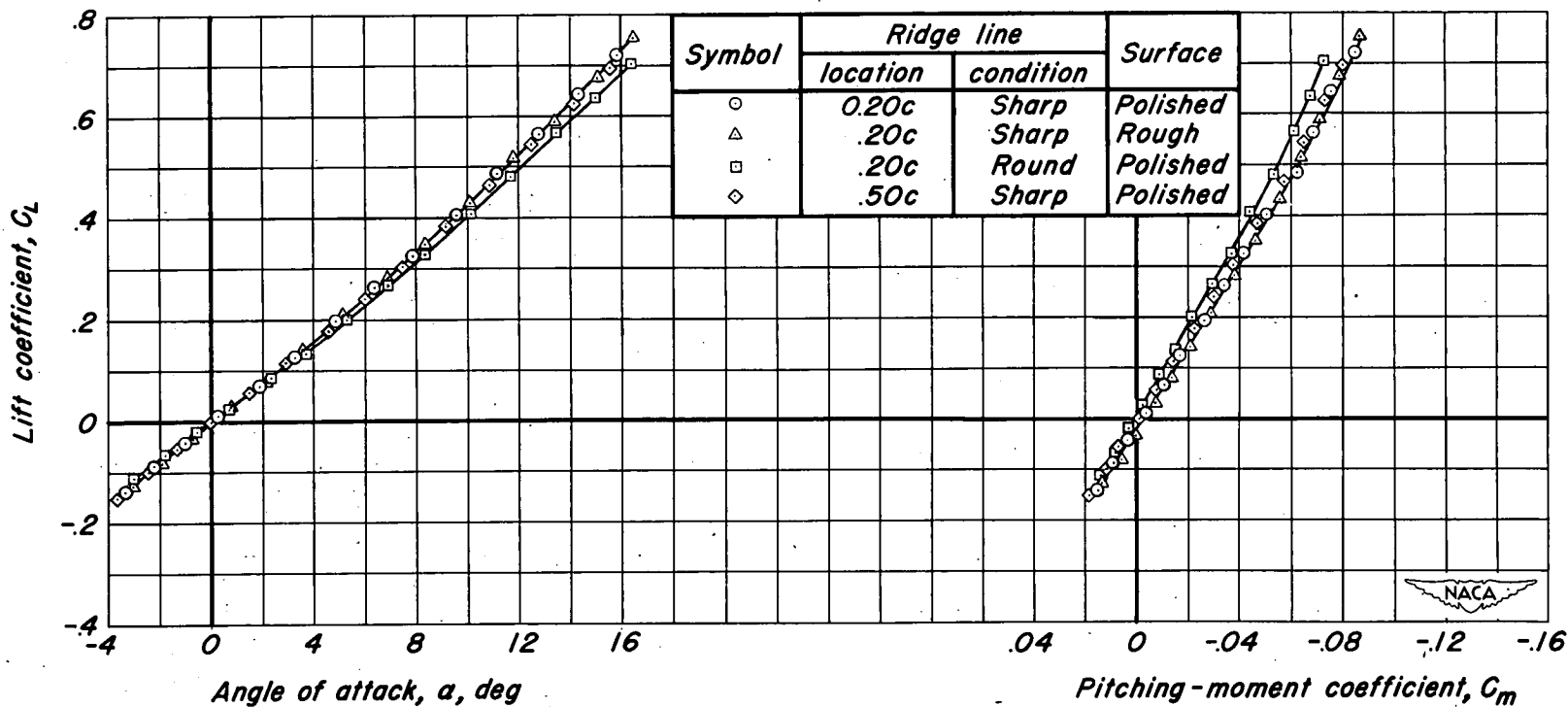
(a) Lift characteristics.

Figure 4.- The characteristics of the triangular wing, maximum thickness at $0.20c$, sharp ridge line, polished surface, no flap.



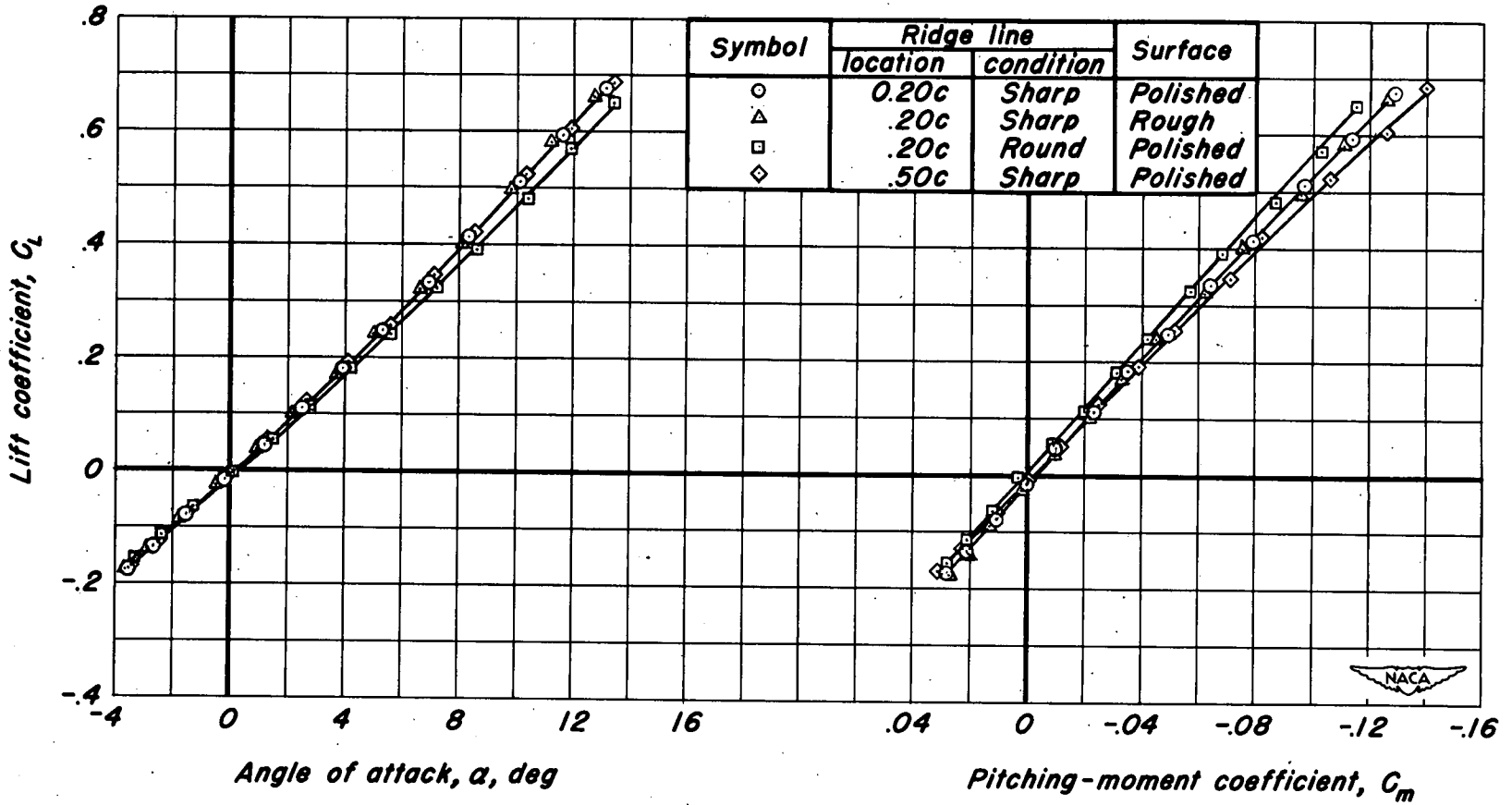
(b) Pitching-moment characteristics.

Figure 4.- Concluded.



(a) $M = 0.47$.

Figure 5.- Effects of airfoil-section and surface-condition modifications at several Mach numbers.



(b) $M = 0.98$.

Figure 5.- Continued .

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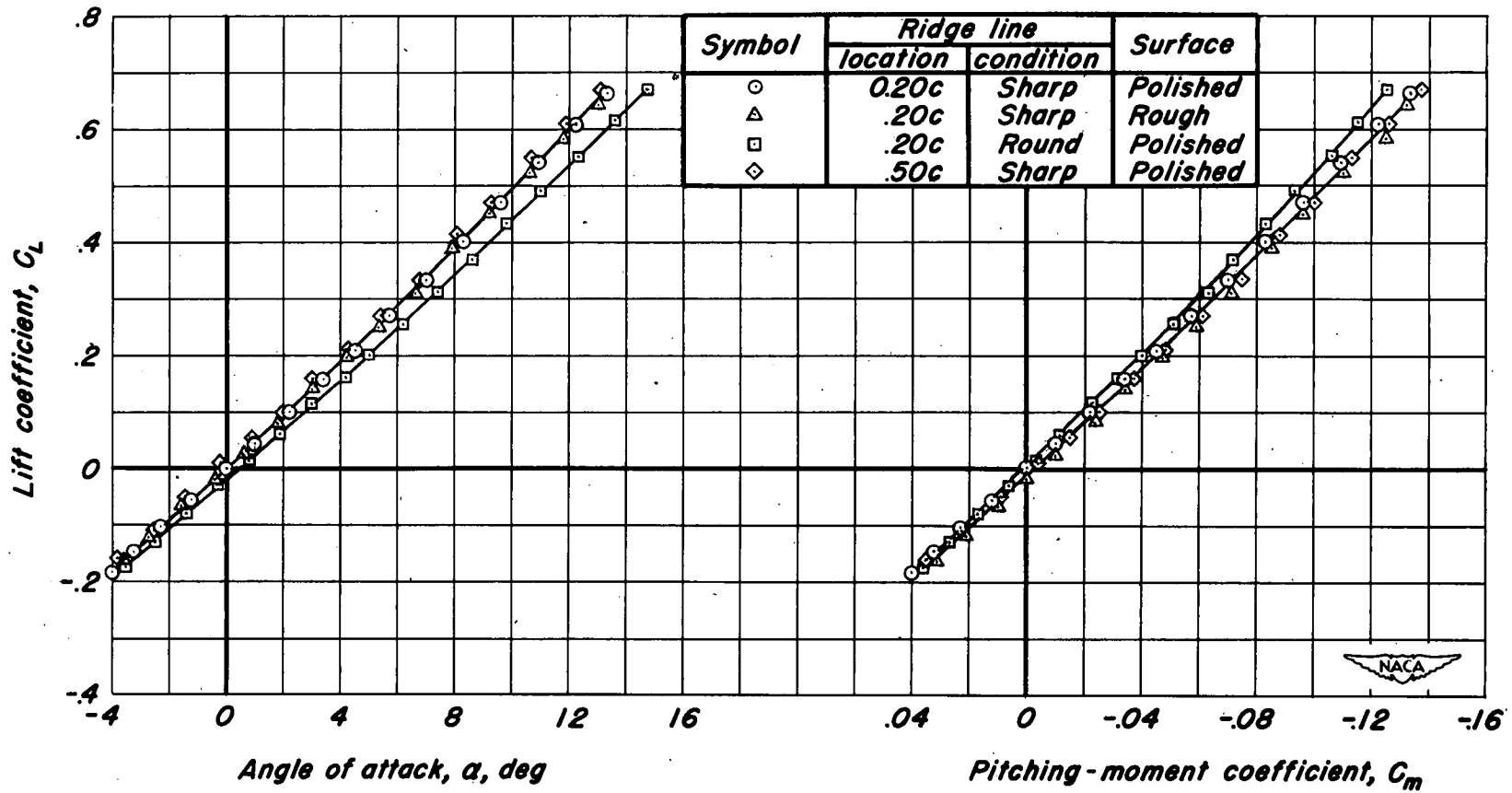
(c) $M = 1.15$.

Figure 5.- Concluded.

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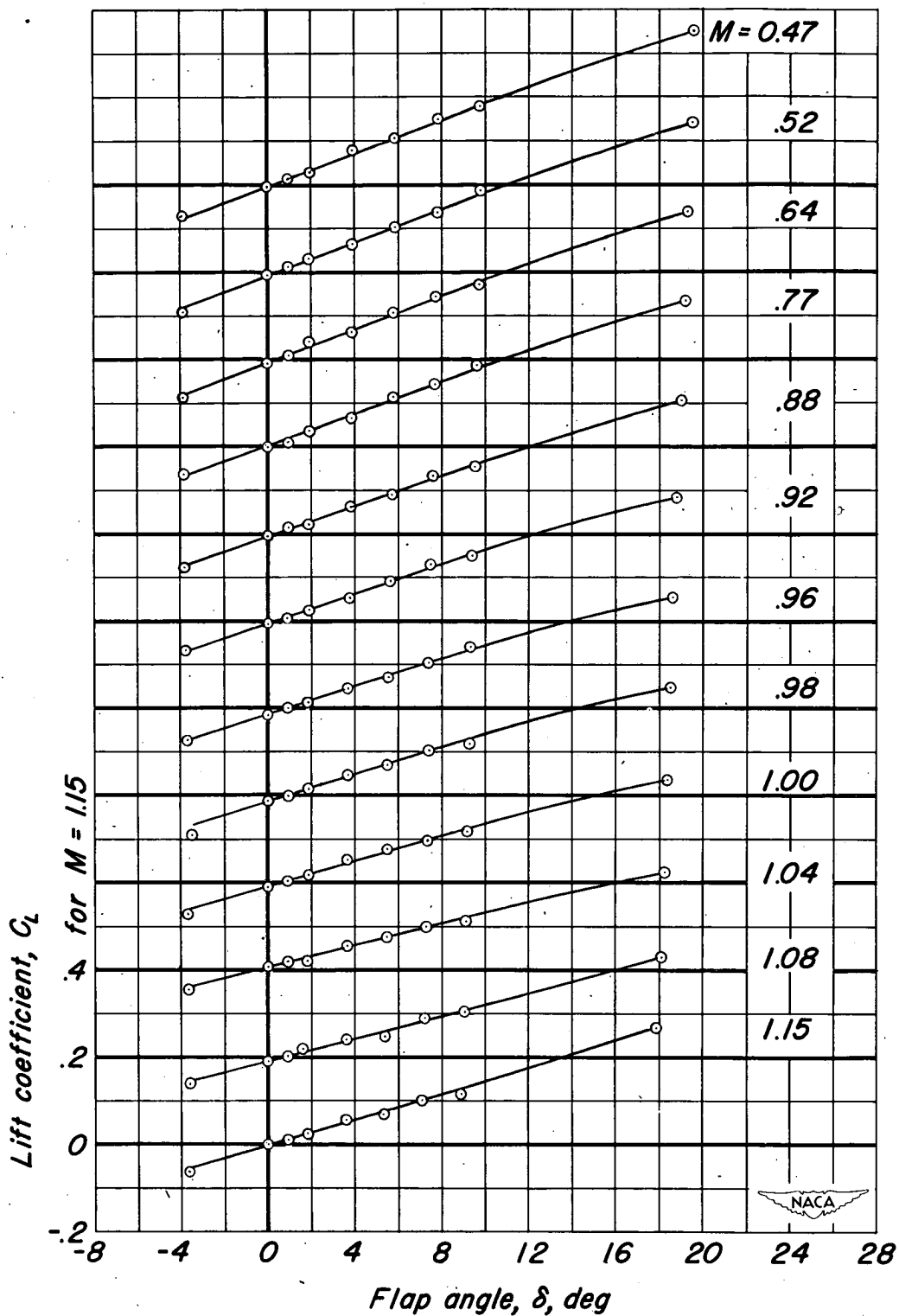


Figure 6.- Characteristics of the constant-chord flap, $\alpha = 0^\circ$.

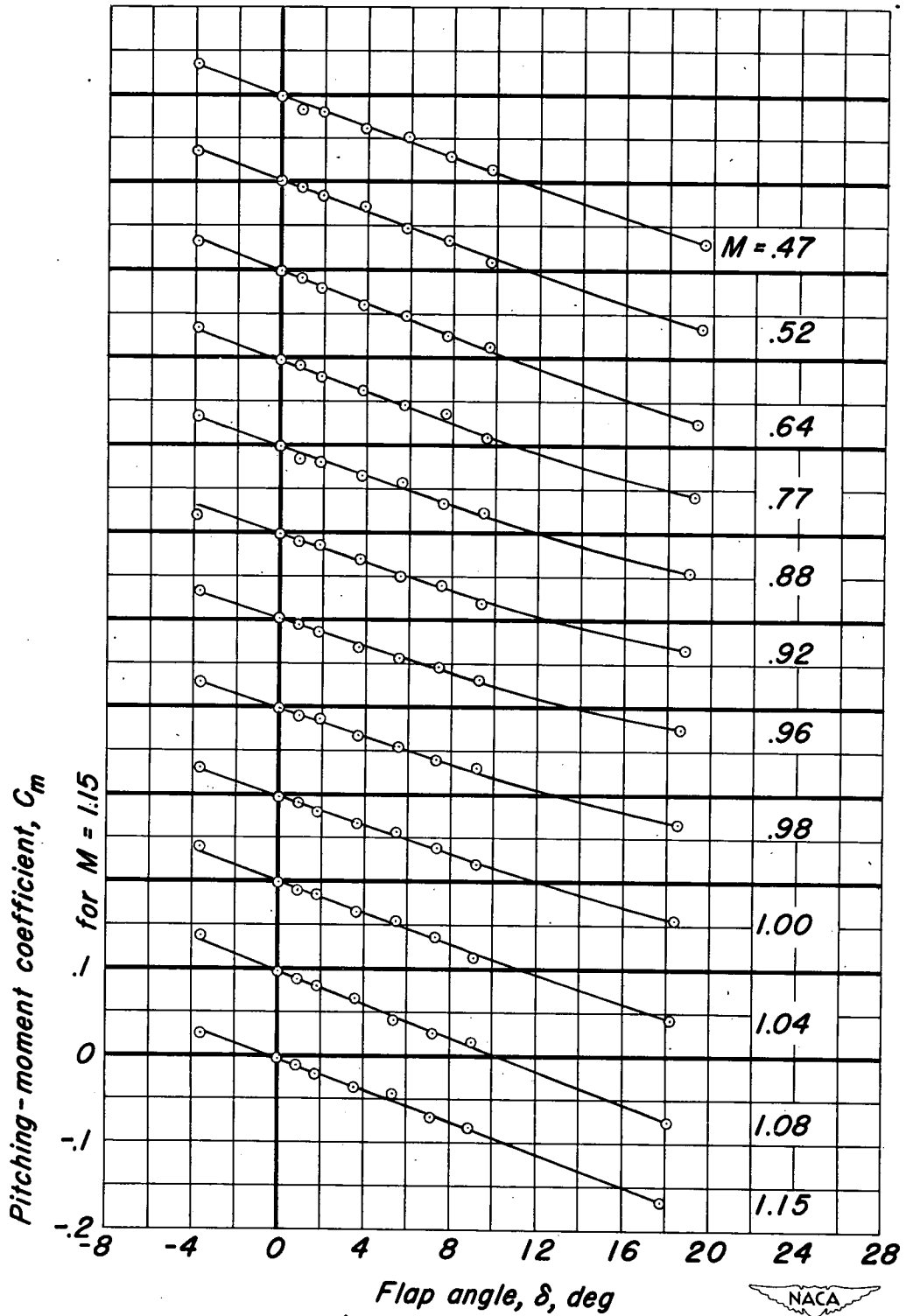


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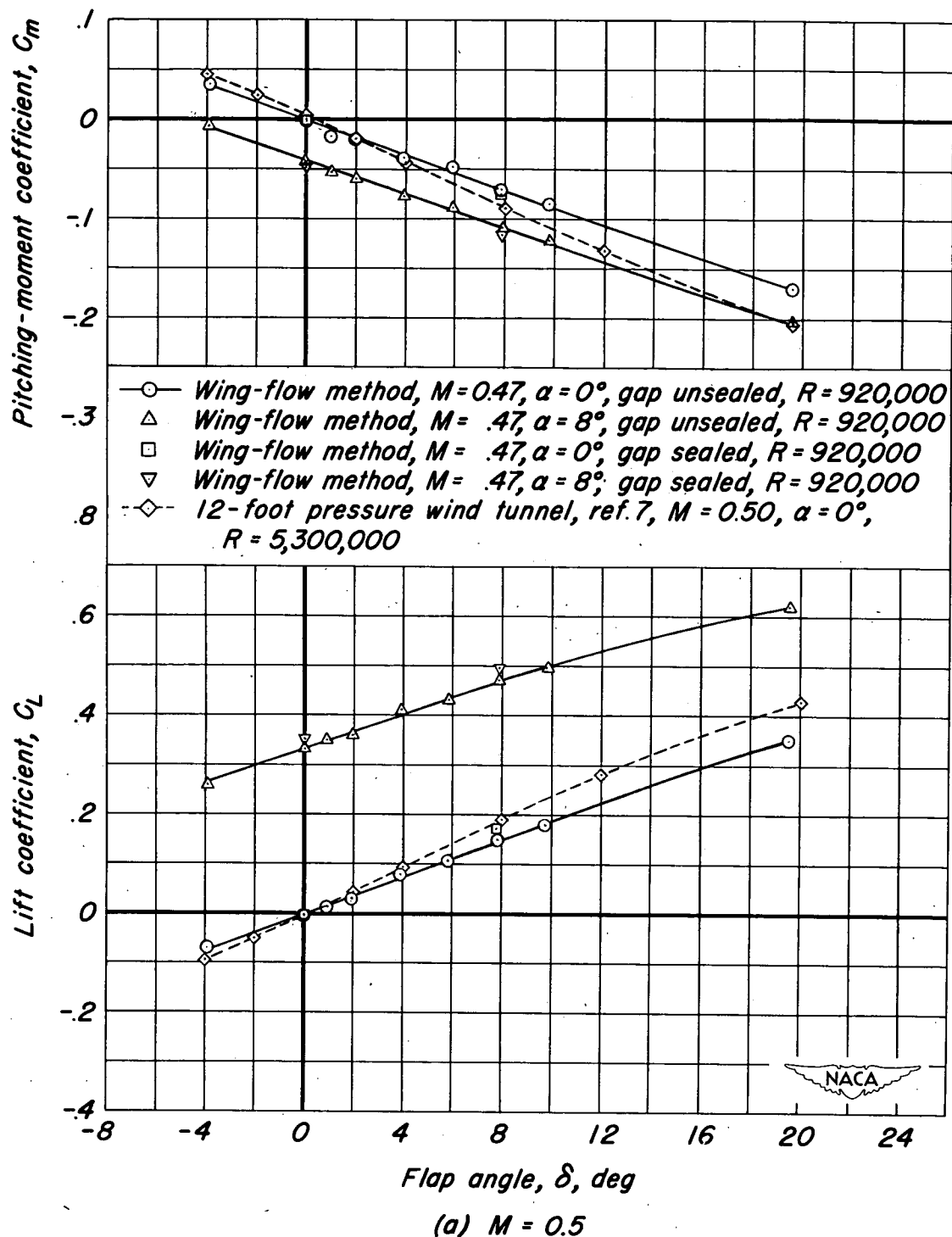


Figure 7.— The effects of changing angle of attack and of sealing the gap between the flap and the wing on the flap characteristics at three representative Mach numbers.

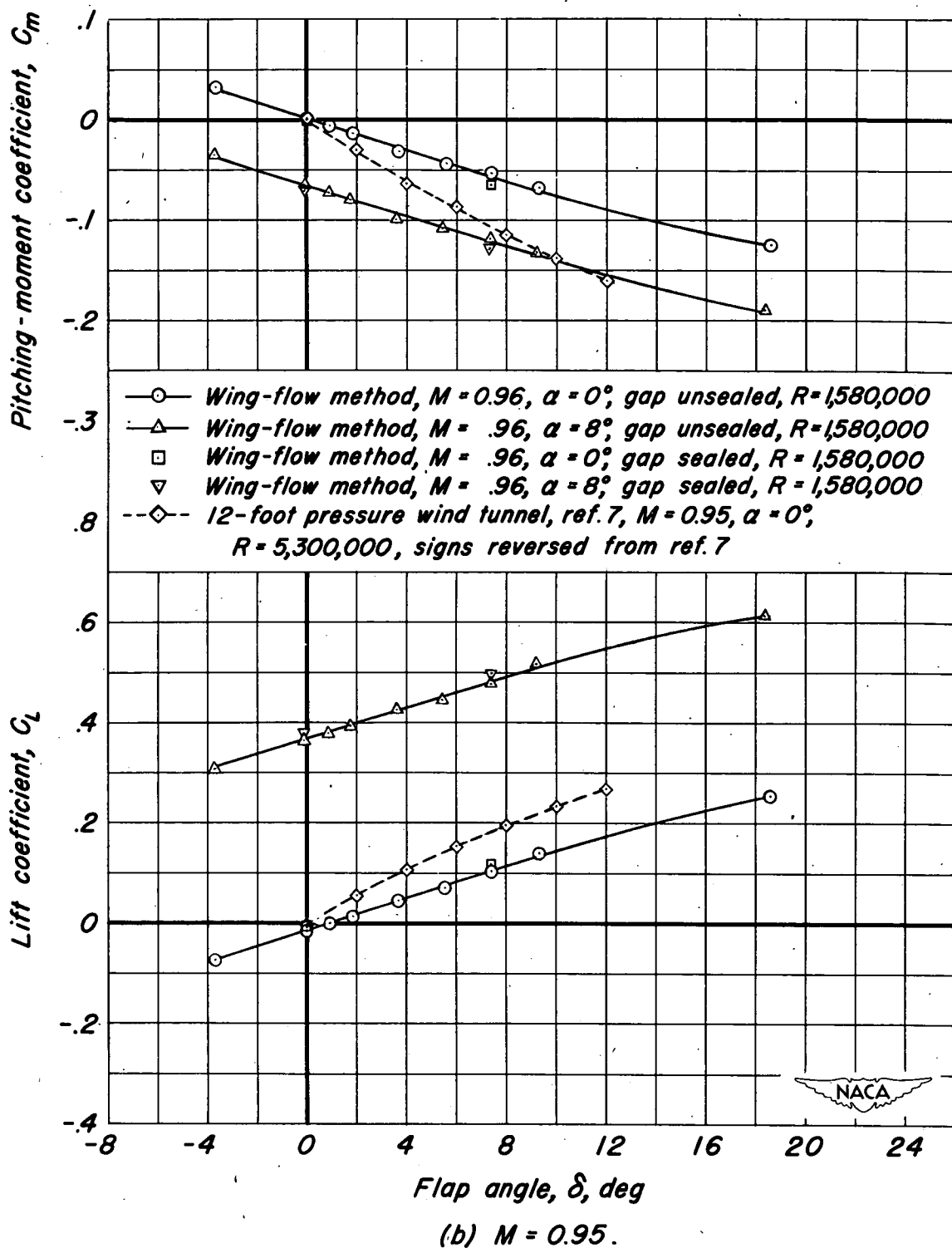


Figure 7.- Continued.

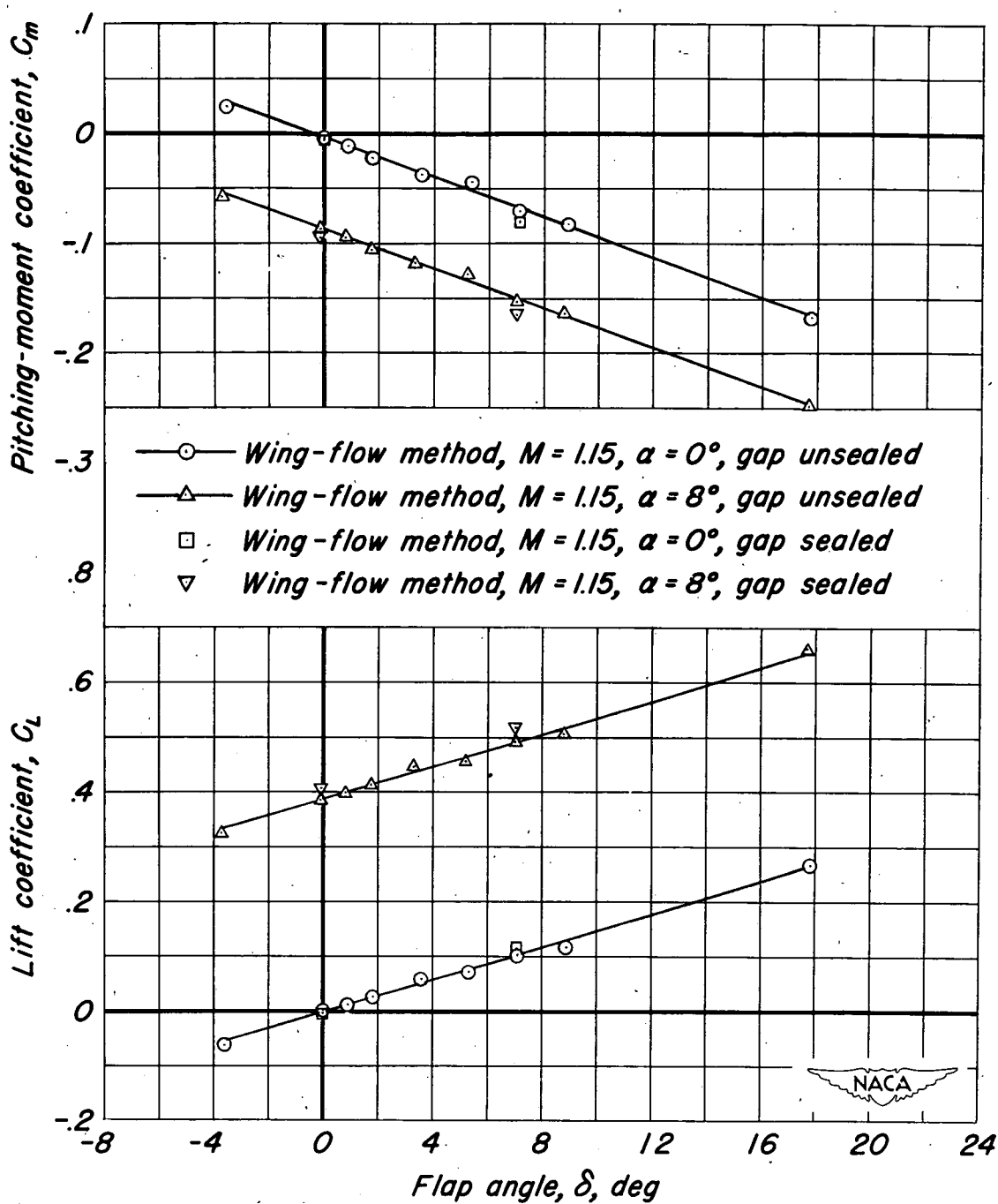
(c) $M = 1.15$.

Figure 7.- Concluded.

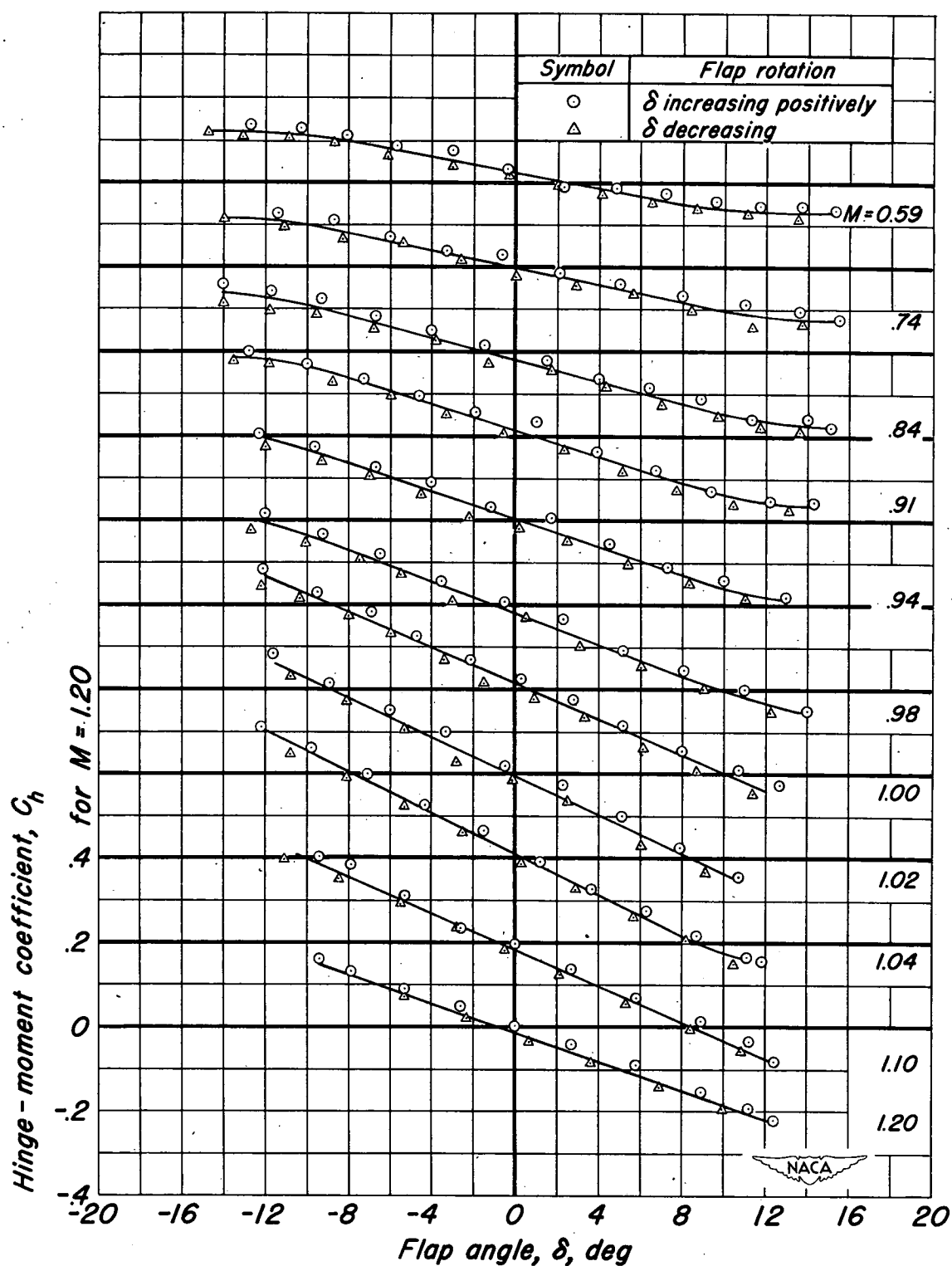


Figure 8.- Variation of hinge-moment coefficient with flap angle, $\alpha = 0^\circ$.

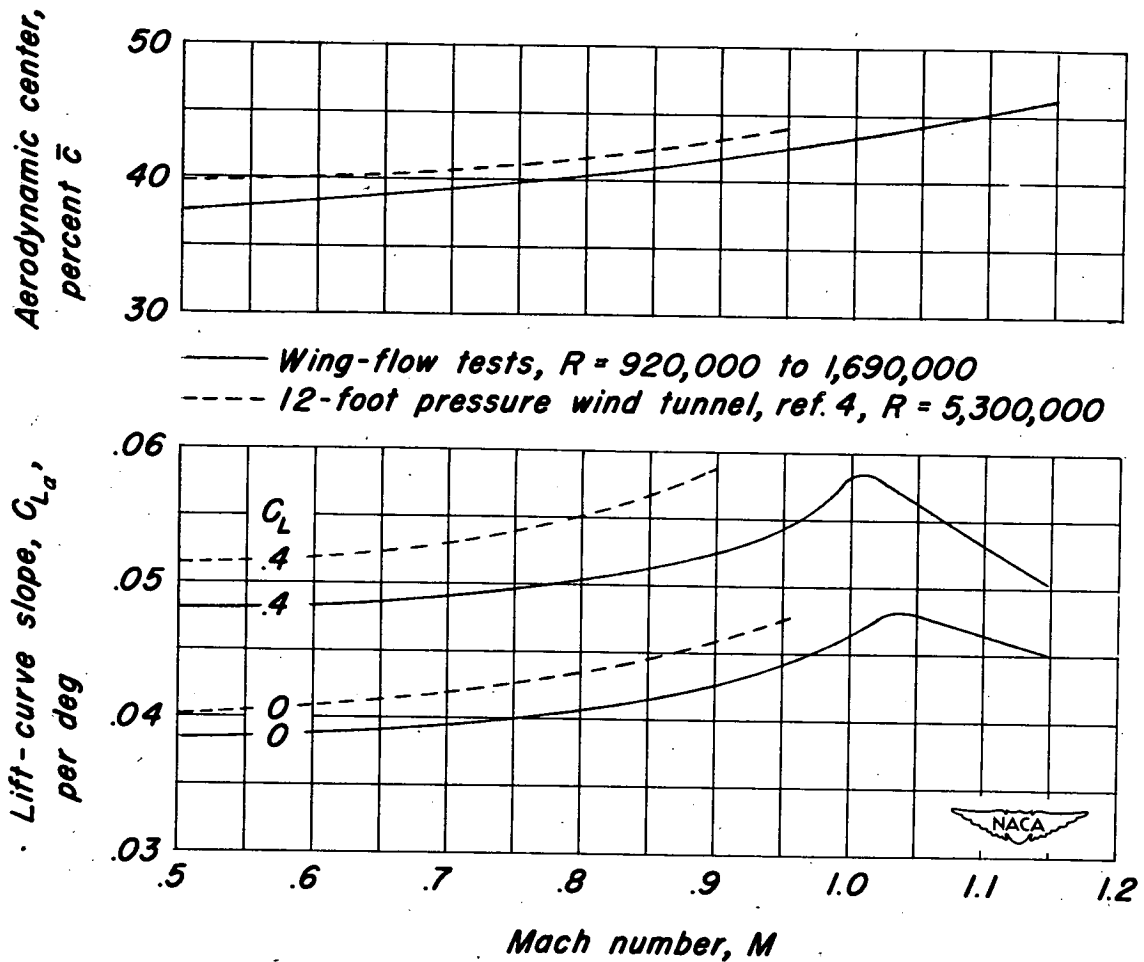


Figure 9.— The effect of Mach number on the lift-curve slope and aerodynamic center position, including comparison with large-scale data. Maximum thickness at $0.20c$, sharp ridge line, polished surface, no flap.

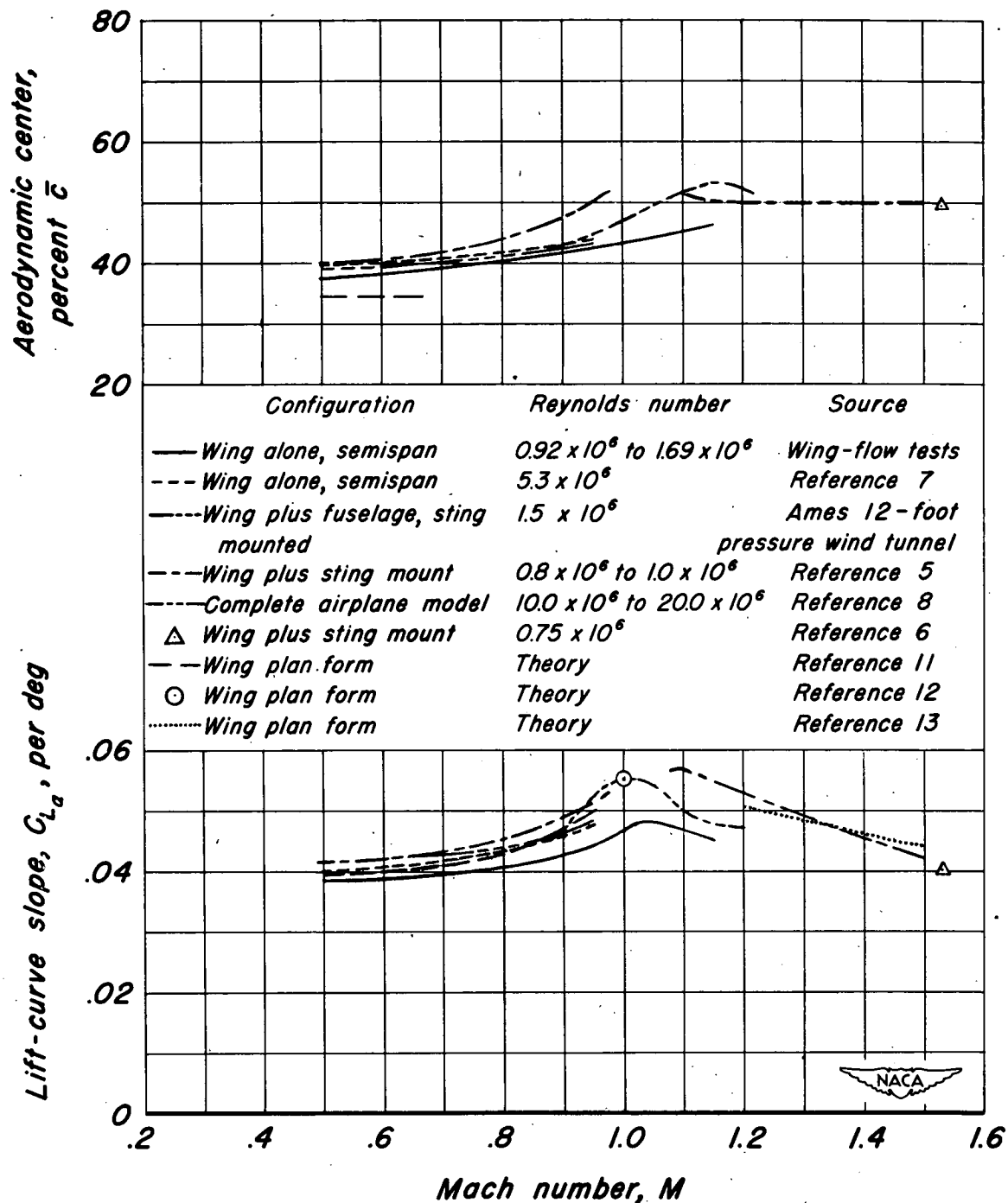


Figure 10. - General comparison of the lift-curve slope and aerodynamic-center location of a triangular wing of aspect ratio 2 as indicated by various tests and theoretical methods.

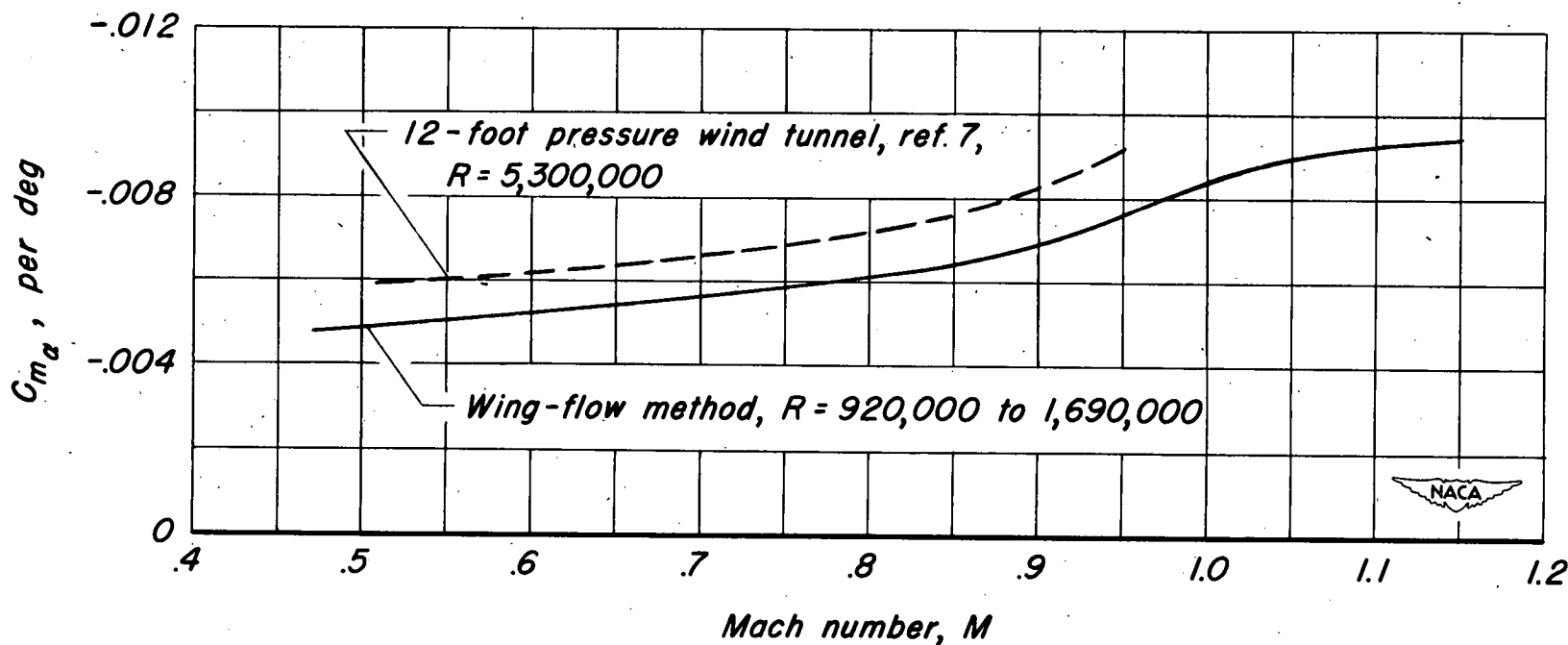


Figure 11.— The effect of Mach number on the variation of pitching-moment coefficient with angle of attack at zero lift for the triangular wing, maximum thickness at 20-percent chord, sharp ridge line, polished surface, no flap.

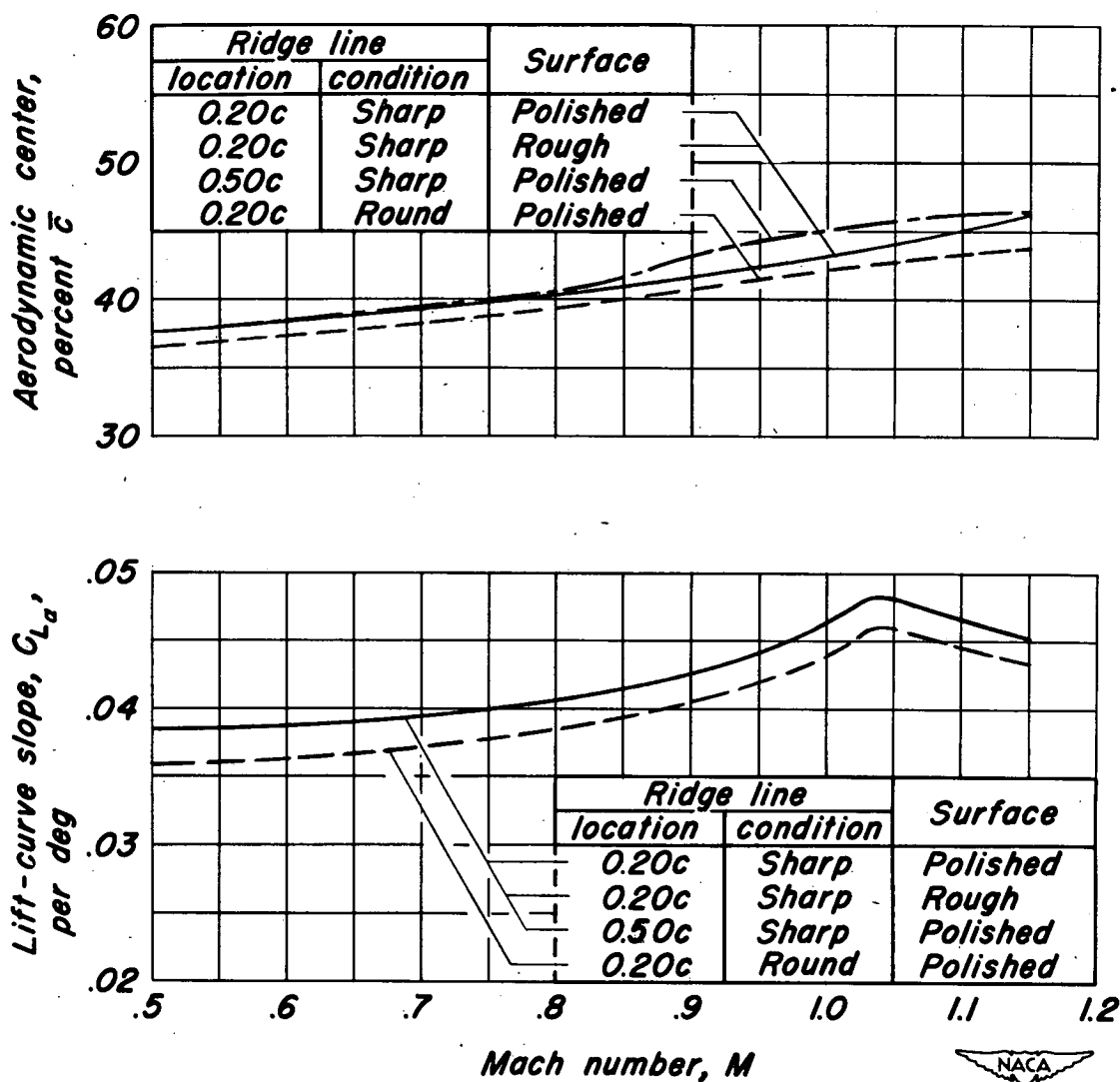


Figure 12.— The effect of airfoil-section and surface-condition modifications on the variation of the lift-curve slope and aerodynamic center with Mach number.

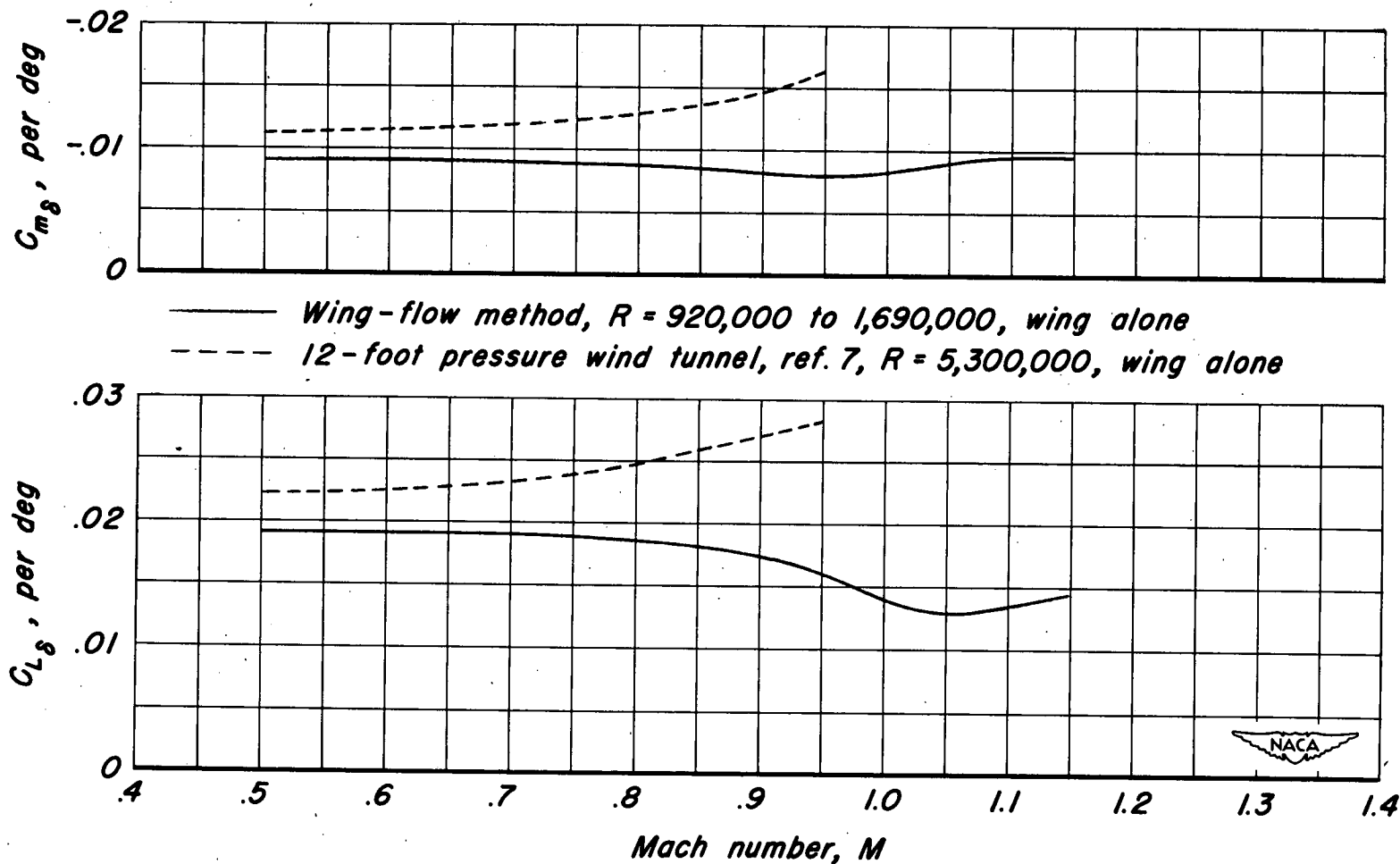


Figure 13.— The effect of Mach number on the lift and pitching-moment effectiveness of the constant-chord flap at an angle of attack of 0° and over a flap-angle range of -4° to $+4^\circ$.

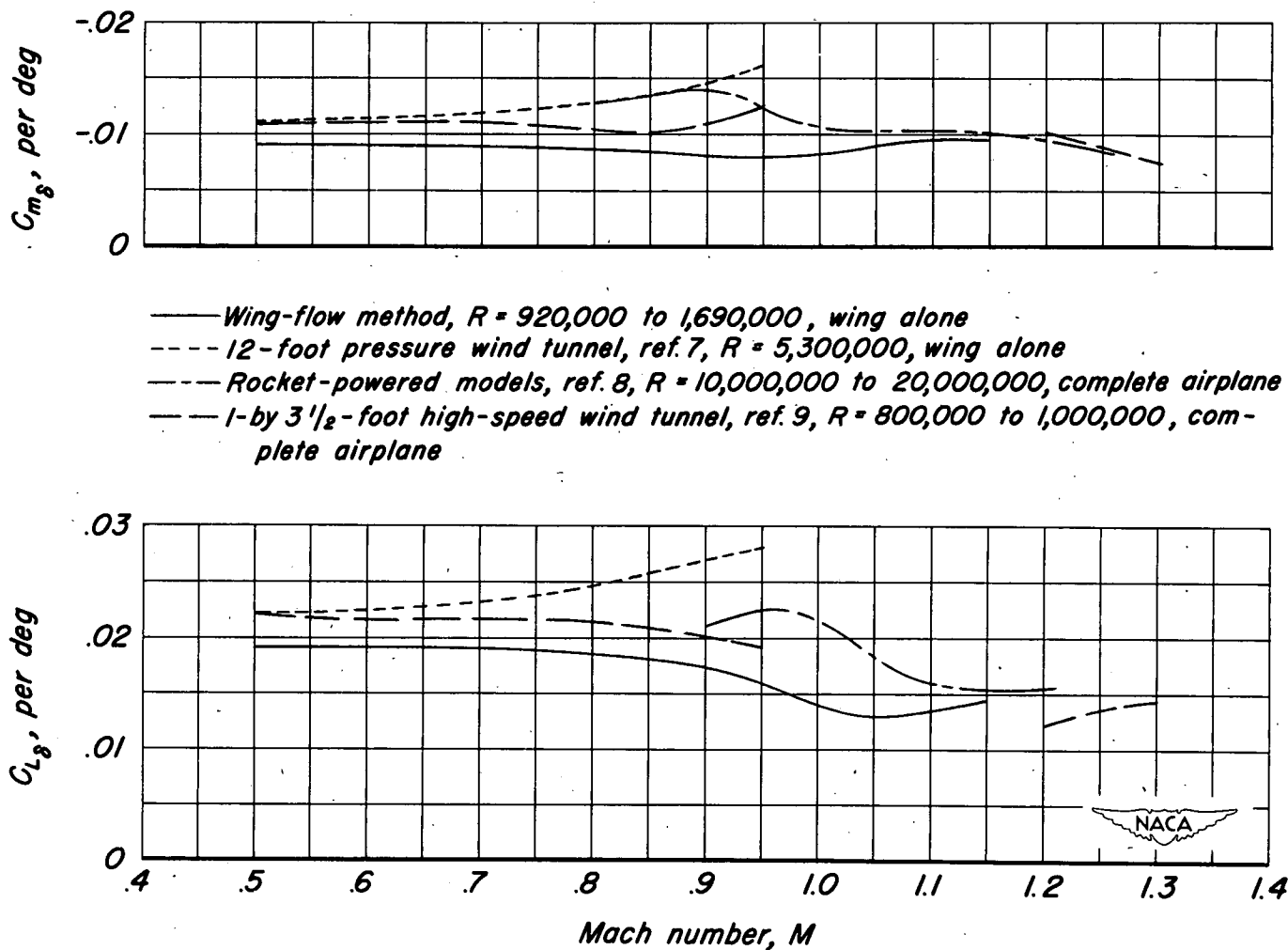


Figure 14.— General comparison of the lift and pitching-moment effectiveness of a constant-chord flap over a flap-angle range of -4° to $+4^\circ$. 0° angle of attack.

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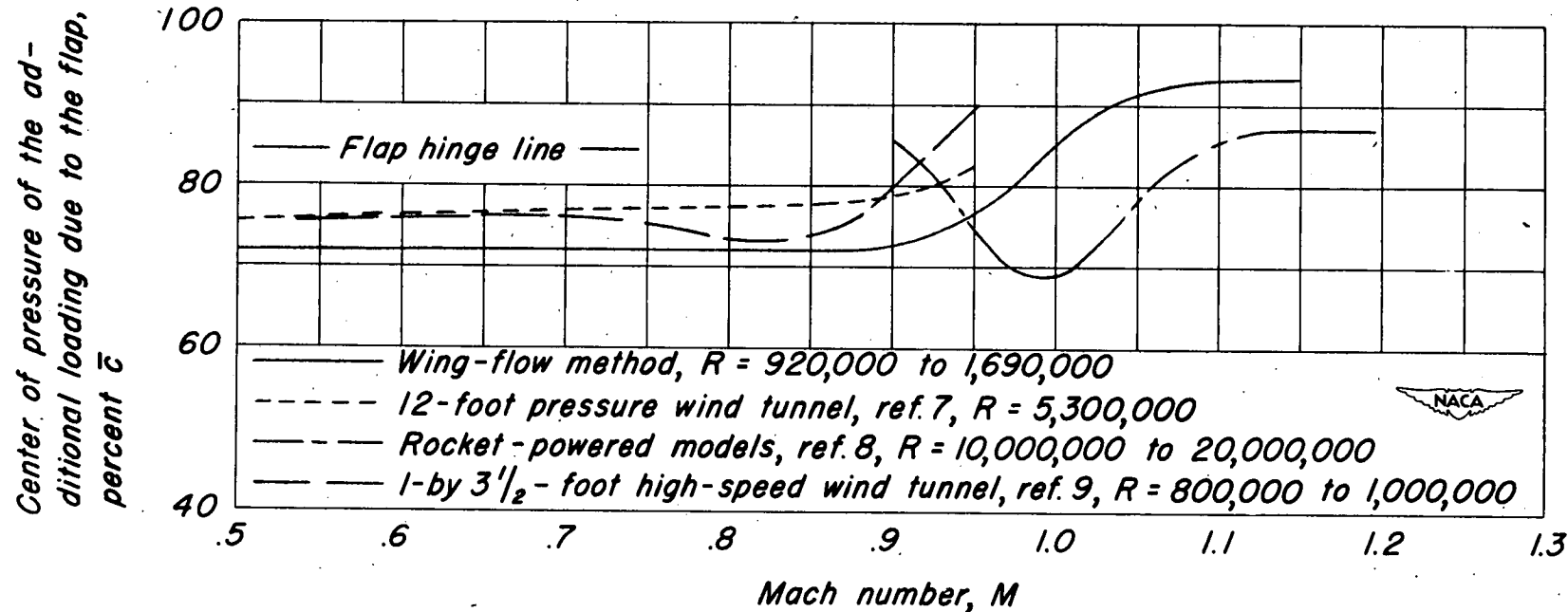


Figure 15.— The effect of Mach number on the center of pressure of the additional loading due to the flap.

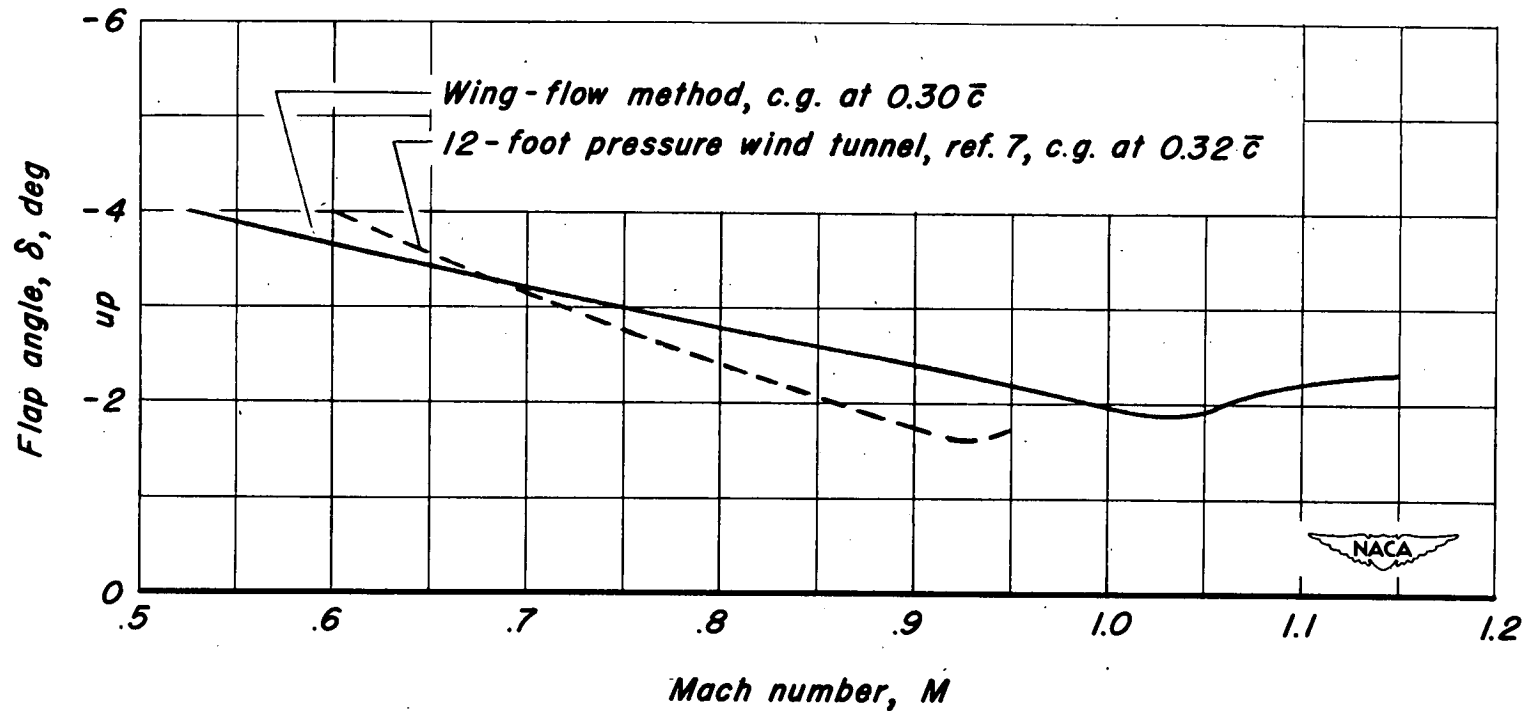


Figure 16.- The variation with Mach number of the flap angle required to balance a hypothetical flying wing airplane in level flight, pressure altitude 30,000 feet, wing loading 60 pounds per square foot, center-of-gravity positions noted.

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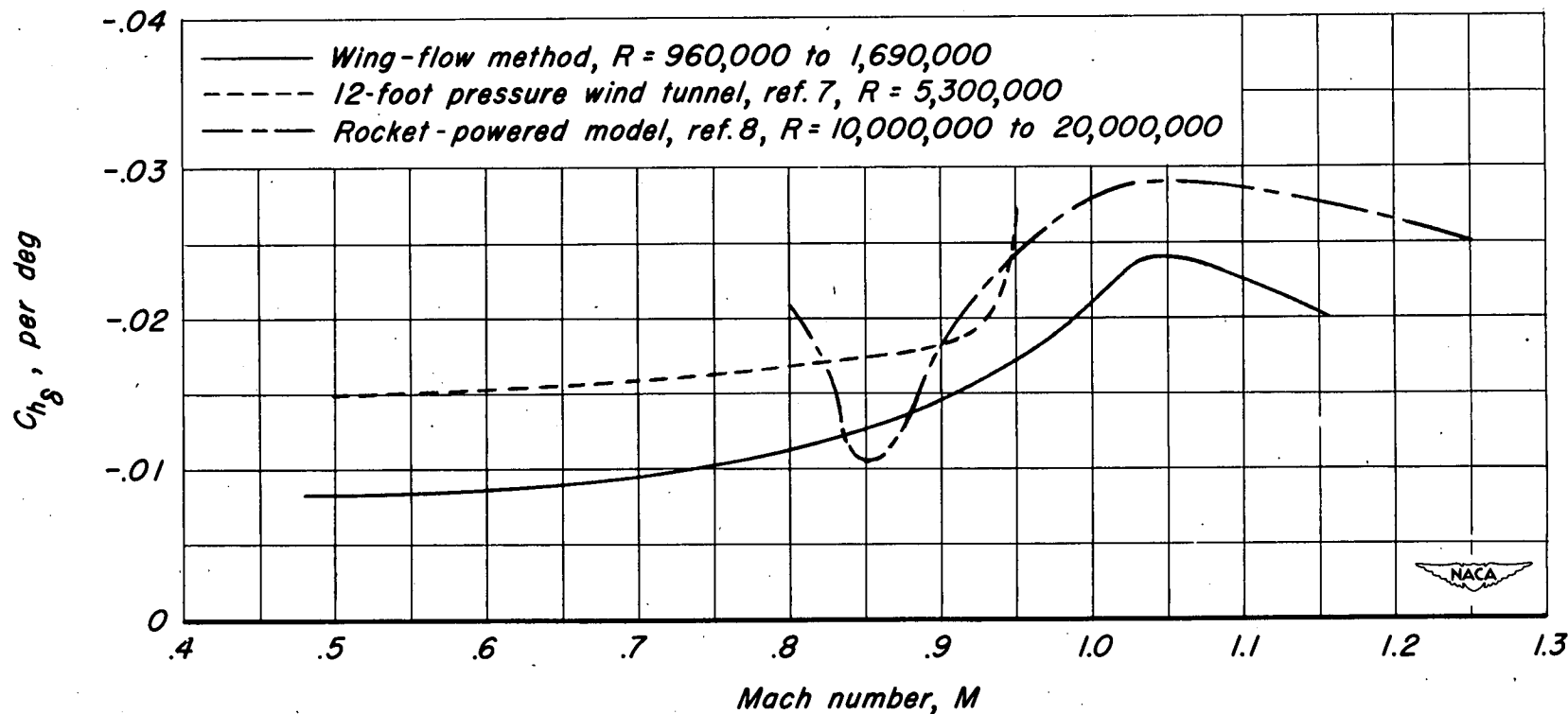


Figure 17.— The effect of Mach number on the variation of hinge-moment coefficient with flap angle at an angle of attack of 0° .